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MPD ARCJET SYSTEM

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Author Introduction

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MPD ARCJET SYSTEM

by

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1. Introduction

In recent years, I have received questions as to the nature
of advances in the development of electric propulsion or
concerning the brightening prospects toward the increasing size
and power of satellites which concern the whole field of
electric propulsion. First, apparently considering Japan's own
strength in this field, a journalist asked "How far have foreign
countries progressed and how is Japan faring?"; also a space
physics technologist wondered "We have the ion engine, the MPD
arcjet, and the teflon thruster, but which is the most
desirable?"; while one student stated "This is naturally very
interesting, but teach me something that will help me in my   /90
studies."

Several years ago, this author also questioned a NASA ion
engine researcher about its future prospects. That person

* Numbers in the margin indicate pagination in the foreign text.

answered my questions, prefacing his remarks with "I, personally, am biased toward the ion engine...", and I, likewise, would like to subtitle this dissertation "Having a Bias Toward the MPD Arcjet" and will attempt to answer the above-mentioned questions from that aspect. Figure 1 is a chart showing objective data which is an overview of all of the active fields of study concerning electric propulsion. This shows the spectrum of articles released at International Electric Propulsion Conferences and breaks down the 80 to 100 articles released at each
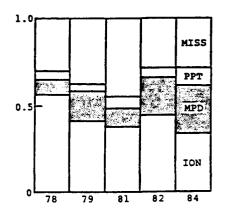

Fig. 1 Changes in announcements of electric propulsion related articles.

conference. You can see, as you look at this, that the number of MPD arcjet related articles has increased with each conference, until recently, when they rank with those about the ion engine. Now, since it is difficult to understand the content of these articles simply from numbers, I will also address their originality. I digress, but I wonder if the reader is aware of the uniqueness of the 1981 data. There were more than the usual number of mission and system related articles, a great many of which were articles from the United States concerning the SEP (solar electric propulsion) mission aimed at a Halley's comet rendezvous. One got the impression that this was quite the most popular subject, but these plans were canceled just prior to that year's conference, in the wake of which other articles were

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withdrawn, making it a rather desolate conference.

The characteristics of the various kinds of thrusters, including chemical rockets, are shown in Figure 2 [1]. This chart plots specific impulse with respect to thrust density, showing the various usage conditions of various thrusters, without expressing the superiority or inferiority of the characteristics. The electric thrusters are dispersed to the right and below the center of the figure. If the
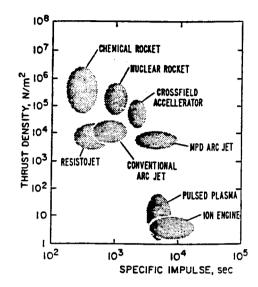

Fig. 2 Specific impulses & thrust densities of various thrusters

size of the thrusters are limited to a certain extent, it can be seen that, among the electric thrusters, the MPD arcjet is affiliated with those in the higher thrust categories. A device called the conventional arcjet is located adjacent to the left of the MPD arcjet in the figure, but this is a relative of the MPD arcjet. The pulsed plasma thruster to the lower right of the MPD arcjet and the teflon thruster, omitted from this figure, also operate very similarly to the MPD arcjet.

2.   Complications in Research and Development

It's a little late to be introducing this, but "MPD arcjet" is an abbreviation of "magnetoplasmadynamic arcjet". Besides this, it is written as MPD thruster or MPD engine, emphasizing

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that it is a thruster. In the following, there will be instances in which it will be abbreviated simply as "MPD", but if the reader would be so kind, I would ask that you read these 3 expressions interchangeably.

Arcjets were discussed as relatives of the MPD arcjet for the purpose of explaining Figure 2, but these have been research since the latter half of the 1950's as jets which used approximately 20,000°K plasma as their source of enthalpy, with the goal giving chemical rockets higher specific impulse. Since the plasma is not parallel to the walls of the discharger at this high a temperature, the amount of heat transferred to the walls becomes extremely great. There is the method of utilizing regeneration refrigeration, as with liquid hydrogen engines, but since increasing the specific impulse tends to reduce flow, it did not ultimately help in cooling the engine, making this method all the more problematic [2]. Consequently, this type of engine did not have sufficient propulsion efficiency or durability. However, its utilization as a plasma torch for welding and cutting did progress on the ground.

The ray of hope that broke through these overcast conditions was a group of American Italians, lead by Gianini, who developed the arcjet under commission from NASA. Through repeated trial-and-error in search of the optimum electrode shape, they announced a shining success in 1964 of a 10,000 second specific impulse at 68% efficiency [3]. The various changes in the discharger shape which accompanied the tests during that struggle can be seen (in Figure 3). The 1950's models were clearly /91

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designed like the chemical
rockets so as to expand the high
temperature plasma in a Ravall
tube, but in the 1960's models,
the dischargers and nozzles were
done away with to try and reduce
the amount of loss to the walls,
eventually becoming a pair of
coaxial electrodes. There was
some concern as to whether
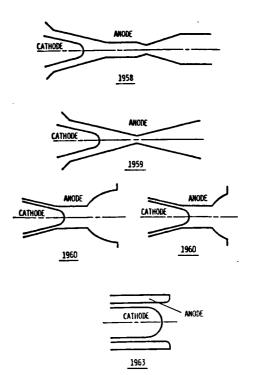effective expansion acceleration
could be achieved in this kind

Fig. 3 Modifications of discharger
configurations in early arcjets

of matrix, but that was answered by a few important secrets which
were discovered by Gianini, et al. [4]. Those secrets were

1) not to use a supersonic nozzle,

2) to reduce the dimensions as much as possible,

3) to keep cooling to a minimum,

4) to work toward homogeneity in the jet cross-section, and

5) to strengthen the electromagnetic effect by supplying a
large electrical current.

In other words, pneumatic acceleration as well as dissociation
and ionization energy had to be sacrificed to compensate for
reductions in wall surface area, which became the compass
pointing toward reliance on electromagnetic energy. The MPD
(magnetoplasmadynamic) arcjet thus came to be so named from these
kinds of complications.

When this report was made, large-scale research was

initiated under an organization consisting of the United States' NASA Lewis Research Center, AVCO, and West Germany's DFVLR. However, as with the best laid plans of mouse and men, data became increasingly undependable as research progressed. Because the knowledge concerning testing environments was lacking and since the performance of the mechanism in the MPD arcjet (as an electric thruster) which exhausted the high-flow propellant was too low at this time, a great deal of testing was performed in high back-pressure chambers without the researchers being aware of the effects of reverse currents on the discharger. Eventually, a surprising phenomenon occurred when impulse was developed with the propellant supply valve closed. It was then discovered, as the result of tests using a sufficiently large vacuum tank, i.e. 4.5 m$\emptyset$ x 20 m, at NASA's Lewis Center, that proper performance could not be obtained unless the pressure was lower than 2 x 10-4 Torr [5],[6]. Thus, the majority of data collected at high back-pressures was rejected including the results in which good performance was obtained. Peak performance thereafter stopped at 2,000 seconds specific impulse at 20% efficiency.

Bad news tends to accumulate in times of bad luck, and in keeping with this, AVCO's Malliaris announced a report to the effect that there was an upper limit to specific impulse [7]. Assigning the variables F to the MPD arcjet's thrust, and $\dot{m}$, V_i, and M_i to the propellant flow amount, the electrolytic voltage, and atomic mass, respectively, and e to the electron charge, the power P required for a plasma state which is near complete

ionization can be considered to be approximately

$$P = F^2/2\dot{m} + \dot{m}eV_i/M_i \qquad (1)$$

When this formula is differentiated with respect to $\dot{m}$ and set equal to zero, namely, by applying the theory of minimum work, this equation becomes

$$F/\dot{m} = (2eV_i/M_i)^{1/2} \qquad (2)$$

and the flow rate, $F/\dot{m}$, becomes tied in with the material values. This effect is the same as the hypothesis put forth by Nobel Prize winner Alfven regarding the origin of the solar system, and is called the "Alfven critical velocity", v_{cr} [8]. Alfven did not work with MPD, but results have been obtained in tests using a circular plasma flow which have shown that, up to the point of complete ionization, the velocity of ionic revolution does not exceed v_{cr} [9]. When this is drawn into the operation of MPD, formula (2) holds true in the state in which the propellant has been completely ionized, meaning that the first and second expressions on the right side of formula (1) have become nearly equal which equates with the achievement of the equipartition of energy. Furthermore, if the input electric power is increased, since the value for F will increase with the electrical power due to electromagnetic acceleration, regardless of the value of $\dot{m}$, the electrical discharge required to balance the equipartition will bind to the electrodes or walls, freely supplying propellant. If argon is used as the propellant, the specific impulse will be stopped at a low v_{cr}=8.7 km/s.

Due to these kinds of circumstances, MPD research became rather discouraging and was dropped by both NASA and DFVLR,

adding to the post-Apollo space recession, and was continued only in the form of academic research at Princeton University in the United States. From another viewpoint, it could also be considered that this situation was due to the fact that the 1960's was not yet the age in which massive electrical power, along the order of kilowatts, was used in space. Arc plasma was used through the 1960's at the Space Sciences Research Lab (now, the Space and Aeronautics Research Center (Uchu Koku Kenkyujo)) in experiments with magneto hydrodynamics, but research on the MPD arcjet was not started until 1970. In contrast to the direct-flow research and development which had been performed up until that time, the testing performed at Princeton University was performed using a quasi-steady pulse 1) with peak maximum current and low back pressure, even in the smaller scale testing, and 2) utilizing the advantage that probe measurement was easy even in high temperature plasmas. We (in Japan) also utilized this system and simultaneously began researching the KOMABA I, a more universal and simple, coaxial electrode MPD arcjet, as shown in Figure 4. Of course, the goal of this was to resolve the critical velocity phenomenon.

As this research advanced, several points of agreement and disagreement with Malliaris' report were discovered that; i.e. that if the discharge flow
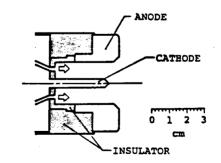

Fig. 4 Discharger configuration in KOMABA I MPD arcjet

was increased, while keeping the propellant flow constant, the discharge became unstable and electrode loss became severe at a

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certain critical point; however, that the velocity at this

critical point was much greater than v_{cr}. Moreover, it became

clear that this critical point could be raised or lowered by the

conditions and method of operation, increasing our ambitions for

improving performance. With the start of the space shuttle and

SEPAC programs in 1976, the MPD arcjet began to be used in space

plasma testing, showing that it was yet incomplete as a thruster,

but providing the opportunity to wrap up testing as a system,

which gave a great push to its research and development.

There was one more great occurrence in 1976 in the field of

electric propulsion. The representative from Russia to the

International Symposium on Electric Propulsion, held in Florida,

announced publicly, for the first time, Russia's space tests and

actual results of test applications in this field [10].

According to this announcement, the Soviet Union had been

performing tests in space using satellites since 1964, and

moreover, that these tests had begun with plasma engines using

MPD as the primary power source, rather than ion engines. Adding

also the data released at the 1978 symposium, the Soviet Union's

applications are as shown in Figure 5. This report was a great

shock to researchers from all of the western countries, and the

Russian representative was barraged with questions, such as "Why

was MPD chosen?" or "What is your future launch schedule?". He

answered, "We will be utilizing MPD to a great extent in high-

powered electric applications or in combination with nuclear-

power for future spacecraft". There are a few unclear places in

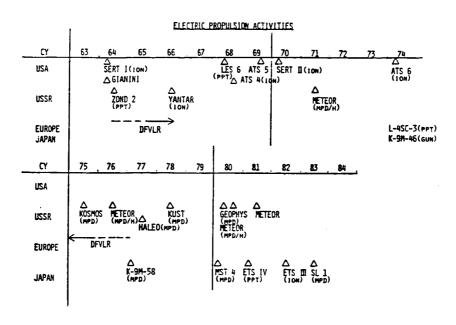the data and figures shown by the Soviet Union and it is

Fig. 5 Space tests and applications of electric propulsion in various countries

difficult to judge actually how accurate they are. However, from the fact that they have reported spraying liquid metals on the electrode surfaces, there are those who believe that the Soviet Union has advanced further than the West from the circumstances which indicate that methods of having circumvented the critical phenomenon.

3. Efforts Toward Improved Performance

At this point, I think that I would like to delve down a little and explain a few of the obstacles which I have encountered during the development of the MPD arcjet. First, I will start by introducing the theory of acceleration as basic knowledge. As shown in Figure 6, a large electrical current flows from a ring-shaped cathode to a coaxial, rod-shaped anode. This large current induces a circumferal magnetic field, which

self-induced magnetic field interferes with the original, large, orthogonal current, passing electromagnetic energy (Lorentz energy) to the plasma. This energy is first transferred to the electrons and the
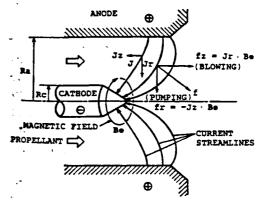

Fig. 6 Thrust generation in MPD arcjet

accompanying ions carried in the electrical current. The <u>/93</u> components of this energy are divided into force in the axial direction, i.e. blowing force, and pumping force, which travels along the central axis. The former is converted directly into thrust, while the latter is first converted into pressure which pushes against the tip surface of the cathode and then becomes thrust. When the discharge current is J and the magnetic permeability is μ, the sum of these is given by

$$F = \mu J^2 b / 4\pi \qquad (3)$$
$$b = \ln(r_a/r_c) + \alpha \qquad (4)$$

[11]. Provided, that the current distribution on the electrode surface is assumed to be uniform. r_a and r_c are the anode radius and cathode radius, respectively, and $a \approx 3/4$ expresses the effect of the cathode tip. As one can see from formula (3), thrust is proportional to J^2, and is determined only by the value for J in a thruster of determinate shape.

In the following, I will focus my discussion on the results obtained at the Space Sciences Research Lab. Most of the discharge testing was performed using the quasi-steady system explained in the previous section. Quasi-steady means that the

operating time is long enough as to seem steady compared to the
flow characteristic time of 2 to 10 µs. The power source used
was a pulse shaping circuit which was able to supply a maximum of
10 kA and the pulse width was 1 ms. The propellant also had to
be injected in quasi-steady pulses in order to match the this
current pulse. In order to accommodate this, a high-speed
electromagnetic valve (fast acting valve, FAV) and a high-speed
ionization vacuum gauge (fast ionization gauge, FIG), to properly
measure the gas pulse, were developed [12-14]. Example
measurements of the gas pulse are shown in Figure 7, but it is

extremely important that the gas
and current pulses be accurately
synchronized. The majority of
the tests were performed in a
1.5 mϕ x 2.8 m vacuum tube,
whose chamber pressure was 10^{-5}
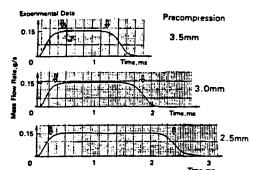Torr prior to operation and 10^{-4}

Fig. 7 Waveforms of gas pulses supplied
from high-speed electromagnetic
valve (FAV), measured while
changing FAV spring pressure

immediately after operation. Thrust was determined by suspending
the discharger and FAV in a pendulum-like configuration and then
deriving the maximum shift in this system after pulse operation.
Besides this, thrust profiles were also taken using an
acceleration gauge.

The thrust-current and voltage-current relationships for the
K I shown in Figure 4 are shown in Figure 8 [15]. It can be seen
that as the current is gradually increased, both of the curves
are in a nearly 1:1 proportion to the current J as long as the
current is small. When the current is small, the majority of the

electrical power is consumed in joule heat, so that the power P is proportional to J^2. Consequently, both the voltage $V = P/J$ and the aerodynamic thrust $\propto P^{1/2}$ become proportional to J. Then, as the current continues to increase, the dependency of the thrust shifts from the lower limit of region 2 in Figure 8
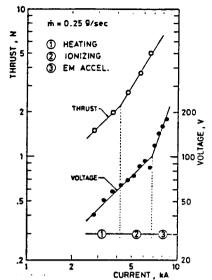

Fig. 8 Thrust and discharge characteristics of K I MPD arc jet using argon operating gas

toward J^2 and the dependency of the voltage shifts from its lower limit in section 3 toward J^3. This fact is a manifestation of the electromagnetic effect, and the fact that the thrust $\propto J^2$, with respect to the voltage, can be understood from the fact that

$$V_\blacksquare = F^2/2\dot{m}J \propto J^3 \qquad (5)$$

is the voltage which is commensurate to the electromagnetic acceleration. Provided, that simultaneous electromagnetic tendencies are not seen in both the thrust and voltage, the fact that J^3 transition in voltage is delayed can be attributed to the fact that electromagnetic acceleration is prevalent during the time that the temperature is increasing slowly and aerodynamic acceleration is plateaued in contrast to the active progression of ionization and joule energy being introduced during region 2. If the lower limit of region 3 is considered to be the point at which the power required for ionization and heating becomes equal to the power required for electromagnetic acceleration, it can be understood that this matches the conditions in formulae (1) and

(2) which induce the Alfven critical velocity v_{cr}. However, it was proven in the testing that, rather than the flow becoming unstable immediately upon entering region 3, a constant flow mass /94 would only become unstable at larger currents. Consequently, the fear that v_{cr} was an upper velocity limit disappeared. In 1976, in an experiment performed in the large-scale space chamber (20 m∅ x 37 m) at NASA's Johnso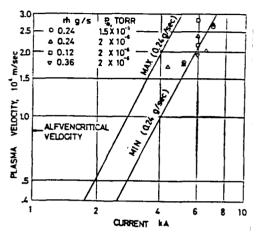n Space Center to demonstrate the SEPAC test equipment, more direct proof was realized. Since the MPD and chamber walls were sufficiently separated, velocity measurement was able to be accomplished by the flight time method without being influenced by particulate reflection from the walls during the operating time [16]. As can be seen in Figure 9, the velocity obtained therein achieved over three times v_{cr}.

Fig. 9 Plasma velocity measurement results in NASA, JSC large-scale space chamber using Ar gas

Because the instability which is supposed to occur immediately upon reaching v_{cr}, according to analysis based on Alfven's hypothesis, did not occur, this theory turned out to be still no more than an unstable proposition; but what other causes could there be?
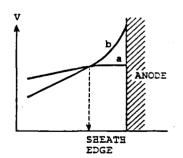
Fig. 10 Oscillation threshold of sheath near cathode

One is a fact that was discovered about vacuum arcs, which is the

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power loss and spot generation in the anode [17]. As shown in Figure 10, the discharge current is increased by stabilizing the plasma density at the edge of the anode sheath (electrostatic boundary layer). Since the current carrying particles (primarily electrons) are relatively reduced when this is done, the sheath (a), which until now had had a slightly negative or homogeneous potential (relative to the sheath edge potential) which repelled the high energy electrons, changes, conversely, to a positive potential, thereby taking on larger quantities of electrons.

When this happens, electron collisions become more severe, eventually forming anode spots and initializing electrode fusion loss. This kind of circumstance must be avoided not only because of the large amounts of power loss in the anode, but also because it also
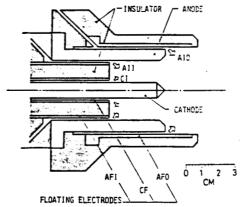

Fig. 11 Discharger configuration in K III MPD arcjet. AII,AIO,&CI each represent propellant sprays from inside and outside of cathode and from anode

shortens the life of the electrodes due to fusion loss. The K III model, shown in Figure 11, was devised as a means of dealing with this problem [18]. A gap was incorporated around the circumferences of the bases of the anode and cathode, enabling the flow mass ratios at their outer edges to be controlled. As a result, the propulsion efficiency reaches its maximum when the majority of the propellant (argon) is sprayed onto the anode surface, as shown in Figure 12. In other words, propellant (or plasma) deficiencies on the anode surface became more

controllable.

The next problem was at
which position along the flow
was it easiest for propellant
deficiencies to occur. Magneto-
hydrodynamic analysis was
extremely helpful in this [19].
The Hall effect and the
ionization process were omitted
by considering only

Fig. 12 Propellant characteristics while
changing propellant injection
distribution, using Ar gas

electromagnetic acceleration with respect to an extremely simple
quasi-linear model (see Figure
13). By making the basic
equation dimensionless, with L
expressing the length of the
thruster and **BO** the magnetic

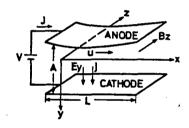**Fig. 13** Acceleration model using linear
MHD channel flow

field intensity at the edge of the flow, the motion equation
becomes

$$\frac{dv}{d\xi}=2R_m\delta\left(\frac{\phi}{\delta}-\frac{vb}{2}\right)b \qquad (6)$$

with respect to the dimensionless velocity, and the induction
equation is given as

$$\frac{db}{d\xi}=-R_m\left(\frac{\phi}{\delta}-\frac{vb}{2}\right) \qquad (7)$$

with respect to the dimensionless magnetic field intensity **b**.
Provided that ξ is the dimensionless coordinate $x=\xi L$,

$$R_m=\sigma B_0^2 L\tilde{A}/\dot{m}, \quad \phi=\mu V\dot{m}/B_0^2\tilde{A}^2 \qquad (8)$$

each express the magnetic Reynolds number and the dimensionless

potential, respectively, in which $\tilde{A}$ is the average electron gap. Also, $\dot{m}$ is the flow mass per unit width in the channel (see Figure 13). If one considers that R_m includes the velocity $B_0^2 \tilde{A}/\mu\dot{m}$, and that

$$B_0 = \mu J \qquad (9)$$

is true, it can be seen that the critical parameter $J2/\dot{m}$ is also included. Boundary conditions are

$$\xi = 0 : b = 1, \quad v = 0$$
$$\xi = 1 : b = 0 \qquad (10)$$

It appears that 3 conditions are given for a 2nd degree equation, but the equation can be solved correctly because ϕ, in other words, the voltage V, is not yet determined. In the case of a constant electrode gap, formulae (6) and (7) are combined and integrated, becoming

$$v = 1 - b^2 \qquad (11)$$

which, when substituted into formula (7), yields

$$\frac{1}{R_m}\left(\frac{db}{d\xi}\right) = -[\phi - (1 - b^2)b/2]$$
$$\equiv F(b ; \phi) \qquad (12)$$

The coefficient F is shown in Figure 14 as the solid line within the domain for b which is defined as $0 \le b \le 1$. When ϕ is reduced, shifting the curve upward to touch the b axis, the integration of formula (12)

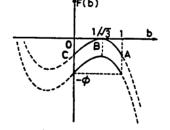Fig. 14 Threshold of coefficient $F(b,\phi)$

starts from the point A on the phase plane of Figure 14, and stops so that $F=0$ at point B. This situation is related to the peculiarity when $R_m \longrightarrow \infty$. The ξ distribution of b when R_m is sufficiently large is shown in Figure 15 and compared to when R_m is small. If R_m is small, the magnetic field is diffused in the

interior of the accelerator and distributed over a nearly straight line, but if R_m is extremely large, $b=1/3^{1/2}$, in other words, $db/d\xi \simeq 0$ at point B in FIgure 14. As R_m becomes larger, a situation is seen in
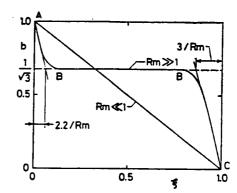

Fig. 15 Dependence of magnetic Reynolds number **Rm** on magnetic field distribution in flow direction

which the magnetic field shaves off in the vicinity of the intake ports and is carried downstream to accumulated near the exhaust. The points **A**, **B**, and **C** in Figure 15 correspond to the same points in Figure 14. Since $db/d\xi$ represents the electrical current, current does not flow near the center of the thruster for the most part, but concentrates near the intake and exhaust ports, flowing within the $O(R_m^{-1})$ boundary layer. This is fortunate for the thruster because the above-mentioned propellant deficiencies and spot formations thus occur only locally. Particularly, in the vicinity of the exhaust port where this is normally excessive due to the the drop in plasma density which occurs as acceleration progresses. Furthermore, when the Hall effect is added, current flows in the axial direction, causing the plasma flow to become separated from the anode surface and worsening the situation [20]. Therefore, the electrodes were formed into the shape of a nozzle as a method of homogenizing the current [21] or as a means of distributing the propellant distribution well, preventing propellant deficiencies [22].

The result of these efforts to date has been to obtain 57% efficiency at 6,000 seconds impulse and a thrust/power ratio of

35 mN/kW at 1,000 to 2,000 seconds specific impulse [22-23]. The keys to thruster design are 1) to perform injection so as not to cause propellant deficiencies on the cathode, 2) to prevent the destruction of the electromagnetic plasma hold on the operating wall, and 3) to select an electrode configuration on which current concentrations have difficulty forming.

4. System Concept

In looking back over the discussion up until now and enumerating the unique characteristics of the revised MPD thruster system, this list would include

1) a large thrust density, which is easily correlated to high energy,

2) simple construction, operation, and application,

3) absolutely no warm up is needed,

4) low operating voltage (charge : 300V, discharge : 150V),

5) little contamination of space ship, space stations if N_2H_4, NH_3, or H_2 is used,

6) abundance of types of propellants which can be used: e.g. N_2H_4, NH_3, H_2, H_2O, CH_4, CO_2, rare gases, etc., and

7) the specific impulse can cover a wide range.

The fact that the operating voltage is low also makes it possible to directly drive the thruster with a solar cell array, simplifying the power rectifier. If the propellant has N or H as its primary molecule, the effects of sputtering are minimal, and because they are elements which were originally existent in the

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surrounding environment, I believe that there is little worry of pollution in the future even if large quantities are used. Also, since N_2H_4 are already in wide use in conventional satellite systems, when combined with MPD, if a compound thrust system which is capable of producing 240 and 2000 seconds specific impulse is used together with only one propellant tank, the breadth of applications can be expanded further. If used in future space stations, H_2 can be easily obtained from the life support system, and if waste gas (CO_2) is used, we can kill two birds with one stone. Thus, the selection of specific thrusts, propellants, and power supplies which are appropriate for the particular mission or an entire system is extremely important to the design of MPD systems.

ACRONYM
FAV: Fast Acting Valve,
CHG: Charger,
CAP/PFN: Capacitor/Pulse Forming Network
IU: Interface Unit

VAP: Vaporizer,
MV : Main Valve

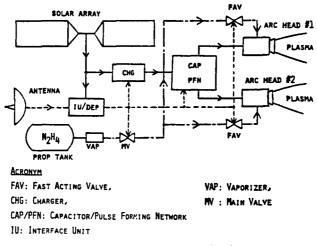Fig. 16 MPD thruster system struture

The structure of a quasi-steady MPD thruster system is shown in Figure 16. The power is supplied from a solar cell array. This output is input to a charger (CHG) at a conventionally established buss voltage, where it is DC/DC converted and charges a capacitor bank (CAP/PFN) in a constant-current system. However, a constant-power charging system is being studied since

the utilization efficiency of the array output is a low 50% in this method [24]. In this system, the solar cells are divided into several blocks, as shown in Figure 17, which blocks are connected in parallel as a high current, low voltage power supply, like **A**, during the first stage of charging, afterwhich, series-parallel switching is performed with **B** and **C**, finally being connected in full series,
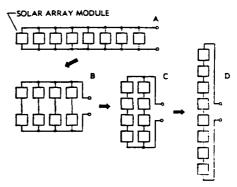

Fig. 17 MPD CAP charging method by series-parallel switching of solar cells

as with **D**, in the final stage of charging, supplying low current, high voltage power. Not only is the output utilization efficiency increased to 80% when this kind of system is used, but the weight of the system is also greatly decreased since the switching elements fulfill the role of a DC/DC converter. The propellant is supplied from a hydrogen storage tank which is also used with the gas jet system, and then stored inside the high-speed electromagnetic valve (FAV) after being gassified and decomposed. Gas pulses are injected into the discharger (HEAD) through the FAV, where they are ionized and accelerated by the current pulses, which are synchronously supplied by the CAP, and finally discharges into the space atmosphere as a plasma jet. The sequence of operation is executed by commands from the processor (DEP), and start and stop commands are executed according to commands received from an earth-based control center by the interface unit (IU).

Next, I will attempt to address the problem of maximizing

the increase in velocity ΔV per unit weight of the MPD propulsion

system. ΔV is derived from

$$\Delta V = (F/M_s)t$$
$$= (F/P)(\Delta EN/M_s) \qquad (13)$$

wherein, M_s is the weight of the space vessel and is considered

constant, t is the total operating time during the mission, ΔE is

the energy accumulated in the CAP, and N is the total number of

shots during the mission. Meanwhile, the breakdown of the mass

of the MPD propulsion system is expressed as

$$M = M_p + M_c + M_h + M_{ch} + M_{cd} \qquad (14)$$

Starting from the first item on the left, M_p is the mass of the

propellant, M_c: capacitor, M_h: discharge head and FAV, M_{ch}:

charger, and M_{cd}: control and interface unit. If τ is the pulse

width and

$$M_p = \dot{m}\tau N \qquad (15)$$
$$\Delta E = JV\tau \qquad (16)$$

are used, the expression

$$\frac{\Delta V}{M} = \left(\frac{F}{P}\right) \bigg/ M_s \left\{ \frac{\dot{m}}{JV} + \left(\frac{M_c}{\Delta E}\right)\frac{1}{N} + \frac{M_0}{\Delta EN} \right\} \qquad (17)$$

is obtained through formulae (13) and (14). Provided that the

capacitor load is expressed as Q,

$$\frac{M_0}{\Delta EN} = \left(\frac{M_h}{QN}\right)\frac{1}{V} + \left(\frac{M_{ch}}{P}\right)\frac{1}{t} + \frac{M_{cd}}{\Delta EN} \qquad (18)$$

is true. F/P is presently about 30 mN/kW, but it is still

expanding. $\dot{m}/JV$ is a quantity which nearly inversely

proportional to the specific impulse I_{sp}, which then of course

decreases as the I_{sp} increases. $M_c/\Delta E$ is the energy accumulation

coefficient for the capacitor, which is 2 to 4 g/J in aluminum

electrolytic capacitor and 20 g/J in metallic depositioned film

capacitors which are presently on the market, which capacitors

are extremely lightweight when electrolytic capacitors are used.

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However, electrolytic capacitors have a short lifetime at N to 104, and when that of the film condensers is considered, i.e. N to 108, the latter is overwhelmingly advantageous to the extent that it is seen in light of $(M_c/\triangle E)/N$. Efforts have been made recently to reduce $M_c/\triangle E$ without reducing N, eliciting estimates of achieving 10 g/J in film capacitors [25-26]. If the allowable limit of electrode fusion loss expressed as $\triangle M_h$, M_h/QN can be written as

$$M_h/QN = (M_h/\Delta M_h)(\Delta M_h/QN) \qquad (19)$$

and with $M_h/\triangle M_h$ to 50 as the standard allowable limits, 0.5 µg/C is the result for $\triangle M_h/QN$ up to now [27]. As can also be seen from the first item on the right-hand side of formula (18), it was discovered that it is advantageous to take a high value for V (and, consequently, a large value for $_J$). Since M_{ch}/P is considered to be a coefficient which is nearly constant, the second item on the right-hand side of formula (18) is advantageous in increasing the value for N. Because M_{cd} is a mass which is independent from the scale, it is ineffective to the extent that it is large. From the above expressions, it appears that the larger the value for N the better, but from the viewpoint of the longevity of the discharge head, the FAV,

Fig. 18 External view of K X MPD arcjet

and the capacitor, this becomes more difficult as N exceeds 107. In tests on systems which contained elements which were considered essential up until now, presently, 106 has been

achieved, and prospects for 10^7 have been obtained, using a thruster like that shown in Figure 18 [26-27].

There also DC systems among the MPD propulsions systems, development of these advancing in West Germany. Since capacitors and FAV's are unnecessary in DC systems, it can be expected that the systems masses will be reduced and their reliability increased. However, since they are unsuccessful if the power supplied is not greater than 50 kW, the time before their application will probably be delayed. Quasi-steady pulse systems can be made corresponding to average powers of from 1 kW, and if the power is increased, the operating efficiency is increased, also enabling those results to, in the future, gradually approach the performance of the DC systems. In addition, the quasi-steady pulse system is also advantageous because throttling is possible with the operating points fixed by changing the pulse frequency and since the fusion loss is minimal per average power when the peak power is increased.
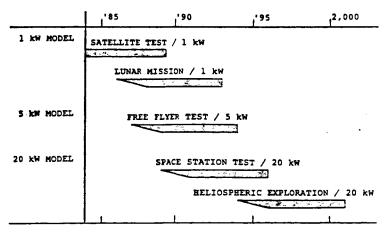
5. Application in Missions

Fig. 19 MPD thruster development plan

Examples of missions in which MPD propulsion systems are expected to be used are shown in Figure 19. 1 kW grade systems will be field tested in satellites in around 1990, and it is anticipated that they will be used in polar orbit investigation missions around the moon sometime during the 1990's [28]. The goals of this mission would be to

1) perform studies near the lunar surface using a space craft would spiral down from a polar orbit around the moon,

2) establish orbital control techniques for placing space craft in high altitude polar orbits around the moon, and to

3) cause a space craft to spiral down to a low polar orbit, using an MPD thruster system, thereby proving its propulsion performance.

An artists concept of the a probe at near 100 km above the lunar surface is shown in Figure 20. The space craft is shown in an attitude of descending by emitting plasma from an MPD thruster in the direction of progress while moving to the left. The orbit is situated just at the border between night and day, and rotates in a state
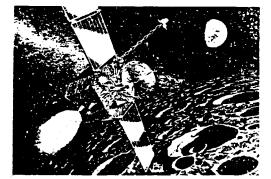

Fig. 20 Proximal exploration of lunar surface by lunar polar orbiting spacecraft using MPD thrusters

of total sunlight, enabling it to receive constant power and continue emissions. Scientific observations which are being considered include the measurement of the distributions of

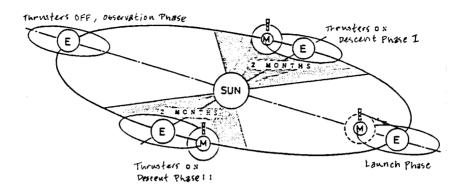magnetism, gravity, and ores, etc. As shown in Figure 21, the

Fig. 21 Application of MPD Thrusters in lunar polar orbit

space craft would be launched to an altitude of approximately

10,000 km, when the earth, moon, and sun are aligned, into an

orbit containing this aligned axis. Approximately two months

after this, the MPD thrusters will be activated in anticipation

of the orbit coming into total sunlight. Two months after that,

as the space craft continues to descend in nearly total sunlight,   /98

it will come to an altitude of 2000 km. In this condition, once

the thrusters are stopped, observations will be carried out for

approximately 4 months while maintaining inertial flight. When

the thrusters are again activated during the next two month

period of total sunlight, the space craft will theoretically

descend to an altitude of zero. Of course, the operating time of

the primary and secondary thrusters will be stopped at random

altitudes so that several observations can be performed during

rotating or while stationary. Also, if the direction of the

thrust is inverted, it becomes possible to move the space craft

from any altitude and return it to a previous altitude.

Consider, for a moment, the propellant required to descend from

an altitude of 10,000 m to the lunar surface and assume that the
mass of the space craft is approximately that shown in Table 1.

Table 1 Table of masses   involved in
lunar polar orbiting probe spacecraft

	Component	Subystem (kg)	Comments
1. Power system		30	
Solar cell panel	20		majority for MPD drive
Battery	5		
Power controller	5		
2. Communications system		10	
3. CDM system		7.5	
4. Orbit-Attitude control system		45	
Gyroscope	2		
Control circuit	10		including foil
Gas jet device	10		
N_2H_4	15		includes MPD propellant
Sensors	7		
Newtation damper	1		
5. Heat control system		10	
6. Structural		25	
7. Observation devices		10	
8. MPD system		27	
Power controller	4		
Capacitor	15		
Discharge head	6		
Propellant supply	2		
Total Mass	**164.5**		

The MPD system comprises 27 kg, but, besides this, of the
majority of the components of the solar cell array and the
propellant, approximately 10 kg are used for MPD operation. The
specifications for the thruster to be used in this mission have
been compiled in Table 2. The number of shots required for
operation at a frequency of 1 Hz is 10 million, but the craft
will have a redundant system equipped with two discharge heads,
as shown in Figure 19, causing them to operate together at only

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Table 2 Specifications for MPD thruster to be used in
lunar polar orbit probe.

Specific impulse	2,000 sec	
Thrust efficiency	30%	
Thermal efficiency	80%	
Thrust/power ratio	30 mN/kW	
Primary input	850 W	* w/ series–parallel
Discharger input	600 W	switching of solar array
Power conversion efficiency	80%	
Pulse width	>600 μsec	
Repeat frequency	≲ 1 Hz	* throttling possible
Propellant	hydrogen or	* composite propellant
	ammonia	system
Allowable repetitions (1 Hz)	10 million pulses	
Thrust	18 mN	
Acceleration	1.2×10^{-4} m/sec^2	
Velocity change	1.2 km/sec.	
(ΔV, 116 days)		

0.5 Hz. The amount of increase in velocity which can be provided
for a space craft which possessing the mass shown in Table 1 by
this MPD thruster is 1.2 km/s, supporting its ability to complete
the mission.

Furthermore, it is believed that a probe mission of the
remote regions of the solar system will be conducted in about the
year 2000 using a 100 kW nuclear powered electric propulsion
system. Outside of the solar system, there are also projections
that MPD propulsion will be utilized with the goal of probing the
plasma of the Galaxy with a flight time of 7 years, shortening
trajectory flight by nearly half. Using hydrogen as the
propellant, the MPD thruster can operate for approximately half a
year at 6000 seconds specific impulse. This kind of specific
impulse has already been achieved at approximately 60% efficiency
[23], and it is anticipated that efforts will be made toward
improving its duration performance as a system.

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The transport of large scale structures from a low-high altitude orbit to a stationary orbit are also being discussed as possible earth orbit missions [29]. In comparisons with chemical rockets, when carrying a 5 ton payload at 50 kW of power, both chemical and MPD systems are rivals, and it is argued that MPD systems are more advantageous in excess of 200 kW with 20 ton payloads. A quasi-stead pulse MPD system was the object of this analysis, but it is believed that DC systems are even better at levels over 100 kW.

6. Conclusion

I wonder if I have made you understand in the above dissertation, but improved performance in the MPD thruster, which has continued in obscurity until its start in the 1970's, has come to progress as has its development as a system. Above all else, this thruster started from the arcjet, and has come to expand the capabilities of the conventional gas jet. Also, because it is a fluid-electromagnetic mechanism, it has piqued our academic interest in magnetohydrodynamics, electrical boundary layers, and plasma turbulence, etc. I request the participation of young researchers who are still burning with the desire to improve performance to become involved with these kinds of problems.

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15. Supplementary Notes

16. Abstract

The current status and future prospects of the magnetoplasmadynamic (MPD) arcjet system are described. Recent research activities on the chemical rocket and electric propulsion are discussed. The characteristics of various MPD arcjet systems including the Komaba-1 system developed by the Institute of Space and Astronautical Science of the Tokyo University are analyzed. The applications of the MPD arcjet system to the satellite test, lunar mission, free flyer test, space station test, and heliospheric exploration are discussed.

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